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INVESTIGATION OF THE BOUNDARY LAYER ABOUT A SYMMETRICAL AIRFOIL IN A WIND TUNNEL OF LOW TURBULENCE

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SUMMARY

An extensive series of boundary-layer surveys was made over the surface of an N.A.C.A. 0012 airfoil at zero lift. The surveys were made at Reynolds Numbers, based on the chord, of 2,675,000, 3,780,000, 5,350,000, and 7,560,000. The drag of the airfoil was measured by the wake-survey method throughout a range of Reynolds Numbers from 225,000 to 7,560,000. The distribution of skin friction over the surface of the airfoil was found from the boundary-layer surveys and the results are compared with those calculated according to the method of Squire and Young developed in England in 1937.

A comparison of the conditions associated with transition in this investigation and those prevailing in previous unpublished tests in the N.A.C.A. 8-foot high-speed tunnel and the N.A.C.A. 19-foot pressure tunnel indicates that the turbulence of the air stream used in the present tests is less than in these tunnels. It appears that the critical Reynolds Number of a sphere cannot be used as a measure of the effects of small amounts of turbulence on the flow about an airfoil.

The distribution of turbulent skin friction calculated according to the method of Squire and Young is in fair agreement with the experimental results. Both theory and experiment show that the skin friction along the surface of the N.A.C.A. 0012 airfoil is approximately 80 percent of the profile drag. The calculated profile drag is in good agreement with that found from the wake surveys.

INTRODUCTION

Because the position and the nature of the transition region on a given body are controlled to a large extent by the turbulence of the air stream in which the tests are

made, boundary-layer determinations about the same body in several tunnels should give an indication of the turbulence of their air streams. Furthermore, if the body chosen for testing is an airfoil, the effect of turbulence will be shown on a body that is applicable to practical aerodynamics.

The flow about the N.A.C.A. 0012 airfoil has been the subject of several investigations. Tests made in the N.A.C.A. full-scale tunnel included the determination of the position of the transition region and the general characteristics of the boundary layer about this airfoil (reference 1). Similar studies, as yet unpublished, of the same airfoil were also carried out in the N.A.C.A. 8-foot high-speed and 19-foot pressure tunnels.

These tests permit a comparison of the effects of turbulence in the air stream in which the present investigation was conducted with those of the air streams in which the previous tests were made. The extensive region of laminar boundary layer in the presence of an adverse pressure gradient that exists on the N.A.C.A. 0012 airfoil affords an excellent opportunity to check experimentally the laminar-boundary-layer theory of reference 2.

The purposes of this investigation are to deduce the relative turbulence characteristics of the low-turbulence tunnel from determinations of boundary-layer transition, to compare experimental with theoretical velocity profiles in the laminar boundary layer, to find experimentally the skin-friction drag and thus to obtain an estimate of the proportion of pressure drag to total drag, and to compare the skin friction calculated according to the method of Squire and Young (reference 3) with that found from boundary-layer determinations.

The tests were carried out on the N.A.C.A. 0012 airfoil set at zero lift. Boundary-layer surveys were made at 12 positions along the surface for each of four Reynolds Numbers ranging from 2,675,000 to 7,560,000. The variation of the position of the transition region with Reynolds Number was determined. The profile drag of the airfoil was determined from wake surveys over a range of Reynolds Numbers extending from 225,000 to 7,560,000.

APPARATUS

The test section of the N.A.C.A. low-turbulence tunnel (reference 4) in which this investigation was made is 3 feet wide, 7-1/2 feet high, and 7-1/2 feet long. It was designed to test airfoil sections in two-dimensional flow. This type of tunnel has the advantage that models can be tested which are much larger compared with the size of the tunnel than is possible with tunnels of more conventional arrangement. In the design of the tunnel, an attempt was made to reduce the turbulence of the air stream to a minimum. Deviations from the desired flow conditions are difficult to determine.

The N.A.C.A. 0012 airfoil on which most of the tests were made had a 5-foot chord and was 3 feet wide, entirely spanning the tunnel. The model was made of wood with a lacquered surface. The lacquer was rubbed smooth with no. 400 water cloth, the strokes running in the direction of flow. Extreme care was used to obtain a smooth and fair surface. A 10-inch-chord metal model was used for determining the drag at low Reynolds Numbers.

A "mouse" was used to make the boundary-layer surveys. The mouse consisted of a group of four total-pressure tubes and one static-pressure tube. The tubes were made of steel hypodermic tubing having an outside diameter of 0.040 inch and a wall thickness of 0.003 inch. The total-pressure tubes were flattened at the ends until the opening at the mouth of the tube was 0.006 inch high. The arrangement was similar to that described by Jones in reference 5. A mouse two-thirds as large as the one just described was used for making measurements particularly close to the surface.

The profile drag of the airfoil was determined with a wake-survey apparatus situated about 24 inches downstream of the trailing edge of the model. The survey apparatus consisted of a movable rake of 25 total-pressure tubes, spaced 0.2 inch, and one static-pressure tube. The total-pressure tubes were connected to a manometer that integrated the loss of total pressure throughout the wake.

$P_S + P_D = P_{TOT}$

$\frac{\rho V^2}{2g}$

METHOD

The velocity distributions in the boundary layer were obtained by measuring the static pressure at a point outside the boundary layer and the total pressure at several positions within the boundary layer. The total pressure outside the boundary layer was used as the reference pressure. From these measurements, the ratio of the velocity in the boundary layer to the free-stream velocity was calculated as

$$\frac{u}{U_o} = \sqrt{\frac{h - p}{q_o}}$$

where u velocity inside boundary layer.

U_o free-stream velocity.

p local static pressure.

h total presure inside boundary layer.

q_o free-stream dynamic pressure.

and y distance perpendicular to the surface.

The heights of the total-pressure tubes above the surface were measured with a micrometer microscope.

The method used in calculating the drag from the wake surveys was essentially the same as Jones' method. The plane of the measurements was sufficiently far removed from the model that, to a first order of accuracy, the drag was given by the area under the curve of total-pressure loss throughout the wake measured by the integrating manometer. The comparatively small corrections to this value of the drag were determined from measurements of the maximum total-pressure loss and the static pressure in the center of the wake.

In order to determine the Reynolds Number corresponding to transition at a given station along the airfoil surface, a function was found which remained constant while the flow in the boundary layer was laminar and which rapidly increased in value with the onset of transition. For convenience, during the tests, this function was taken as

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$$\frac{\sqrt[3]{h_1 - p}}{\sqrt{H - p}}$$

where

h_1 total pressure measured by tube in contact with surface.

H total pressure outside boundary layer.

This function was plotted against $\sqrt{H - p}$. Transition was taken as the point corresponding to the knee of the curve. This procedure is equivalent to plotting

$$\frac{u_1/U_0}{\frac{y_1}{c}\sqrt{R}}$$

as a function of R

where

u_1 velocity measured by tube in contact with surface.

y_1 effective height of total-pressure tube from surface.

c chord of airfoil.

R Reynolds Number based on chord of airfoil.

If y is sufficiently small that the curve of u against y is substantially straight, this function should remain constant at a given station for various values of R as long as the boundary layer remains laminar and should rise rapidly as turbulence occurs, owing to the rapid increase of velocity near the surface.

TESTS

In order to insure that the airfoil was at zero lift, the pressures were measured at two corresponding stations on the upper and the lower surfaces of the airfoil and the angle of attack was adjusted so that these corresponding

pressures were the same. Pressure measurements were then made with a mouse at 57, 54, 48, 42, 39, 36, 30, 24, 18, 12, 6, and 3 inches along the surface from the trailing edge at air speeds of 90, 127, 180, and 255 feet per second.

Boundary-layer surveys were made simultaneously with the pressure measurements. The position of transition as a function of Reynolds Number was determined throughout the range of the test.

The drag was measured before and after the series of boundary-layer determinations.

RESULTS

The pressures are given in terms of the nondimensional coefficient

$$S = \frac{H - p}{q_o} = \left(\frac{U}{U_o}\right)^2$$

where U is the velocity outside the boundary layer at the station in question. The coefficient S is plotted in figure 1 against s/c, where s is the longitudinal distance along the surface from the leading edge of the airfoil.

The curve of section profile-drag coefficient against Reynolds Number for the 5-foot-chord model is given in figure 2. The results given in figures 1 and 2 have not been corrected for tunnel-wall interference.

In figure 3 are presented the values of section profile-drag coefficient corrected for tunnel-wall interference for the 10-inch-chord and the 5-foot-chord models. The corrected drag of the 5-foot-chord model is 5 percent less than the measured drag. The correction for the 10-inch-chord model is negligible.

The results of the surveys of the laminar boundary layer are given in figures 4 to 10. The velocity profiles have been plotted in the form u/U_o against $\frac{y}{c}\sqrt{R}$. Curves

of this type are independent of the Reynolds Number as long as the boundary layer remains laminar. The results for the turbulent boundary layer are given in figures 11 to 18. In these figures, the profiles are given in the form of curves of u/U_0 against y/c. Figure 19 gives the same data plotted logarithmically that are given in figures 11 to 18.

The curves of $\frac{u_1}{U_0}\Big/\frac{y_1}{c}\sqrt{R}$ against Reynolds Number for determining transition are given in figure 20 for several positions along the airfoil surface. Transition is considered to occur at the Reynolds Number indicated by the arrow.

When pressure measurements are made in a region of rapid change of total pressure, the effective center of the total-pressure tubes does not coincide with their geometrical center. When the total-pressure tubes were not in contact with the surface, the effective center of the tubes was taken as the distance from the surface to the geometrical center plus one-quarter the diameter of the tube in the direction of the pressure gradient. (See reference 6.)

It was noticed that the foregoing correction was insufficient to make the readings of the tube in contact with the surface consistent with the readings of the other tubes. The necessary correction to the tube height was found to be a function of $d^2\alpha/\nu$,

where

d height of tube.

α velocity gradient at surface.

ν kinematic viscosity of air.

The trend of the curve of d'/d, where d' is the effective height of the tube, against $d^2\alpha/\nu$ (fig. 21) is similar to that for the Stanton-type surface tube (reference 7).

DISCUSSION

Pressure Distribution

The comparison between the experimental and the theoretical pressure distributions over the N.A.C.A. 0012 airfoil, given in figure 1, shows that the experimentally determined pressures are about $0.05q_o$ lower than the theoretical pressures. Calculations of the effects of the tunnel walls show that the first-order effect is to cause an apparent increase of 2-1/2 percent in the tunnel speed for a 5-foot-chord airfoil of 12-percent thickness. The discrepancy between the calculated and the experimental pressure distributions is accounted for, almost exactly, by the calculated effects of the tunnel walls.

Laminar Boundary Layer

The position of the laminar separation point was calculated by the method of reference 8. The calculations were based on the experimental pressure distribution. Laminar boundary-layer profiles were calculated according to the method of reference 2 at 0.22c, 0.32c, 0.37c, 0.42c, and 0.52c along the surface from the leading edge. These profiles are included in figures 6 to 10. The same approximation to the pressure distribution over the airfoil was used in calculating these profiles as was involved in calculating the position of the laminar separation point.

The shape of the profiles ahead of the position of minimum pressure (figs. 4 and 5) agrees well with the Blasius distribution. The laminar profiles throughout the region of adverse pressure gradient are in satisfactory agreement with the Von Kármán-Millikan laminar-boundary-layer theory. The results of the surveys in the laminar region give no indications of partial transition such as were found by Dryden (reference 9).

Transition

The variation of the position of the transition point with Reynolds Number is shown in figure 22. The development of transition at several stations is shown in figures 11, 12, and 13.

A comparison of the results of this investigation with the results from other tunnels is shown in figure 23. These results are presented in the form of plots of Jones' parameter N against $r = \frac{s_T - s_m}{s_s - s_m}$

where

s_T distance along surface from leading edge to transition point.

s_m distance along surface from leading edge to minimum pressure point.

s_s distance along surface from leading edge to laminar separation point.

The parameter N represents the Reynolds Number of the flow along a flat plate that corresponds to the same value of the boundary-layer Reynolds Number R_δ as the critical value obtained in the test. The value of y corresponding to $\frac{u}{U} = 0.707$ was taken as δ. The value of N was computed as $\left(\frac{R_\delta}{2.3}\right)^2$ where $R_\delta = \frac{U\delta}{\nu}$. No complete boundary-layer profiles were measured on the smooth N.A.C.A. 0012 airfoil in the 8-foot high-speed tunnel, and too few points on the boundary-layer profiles were available in the data of the 19-foot pressure and the full-scale tunnels for an accurate evaluation of δ. Because the theoretical and the experimental values of δ were in substantial agreement in the low-turbulence tunnel and, in order to make the values of N from the various tunnels comparable, the value of δ for figure 23 for each of the four tunnels was obtained from the theoretical boundary-layer profiles calculated by the method of reference 2.

A considerable difference exists between the results of the tests in the other tunnels and the results of the present tests in the low-turbulence tunnel. The results of tests of the same airfoil in the full-scale tunnel show particularly low values of N. Drag tests of spheres in the full-scale and the 8-foot high-speed tunnels (references 10 and 11) show that the critical Reynolds Number of a sphere in the full-scale tunnel is 350,000 as compared with 385,000 for the 8-foot high-speed tunnel and flight. It

is seen, therefore, that neither the difference between the results from the full-scale tunnel and the 8-foot high-speed tunnel nor the difference between the tests in the 8-foot high-speed tunnel and the present tests in the low-turbulence tunnel can be accounted for on the basis of an effective Reynolds Number factor such as was found for sphere-drag tests and the maximum lift of airfoils. A comparison of the results from the low-turbulence and the 8-foot high-speed tunnels indicates that amounts of turbulence in the air stream too small to have any noticeable effect on the critical Reynolds Number of a sphere may have an effect on the occurrence of transition on airfoils. The higher values of N found in the present tests compared with those obtained in the 8-foot high-speed and the 19-foot pressure tunnels indicate that the level of turbulence in these tunnels is greater than the level in the low-turbulence tunnel. Comparative measurements of transition on another airfoil in flight and in the low-turbulence tunnel indicate that the remaining turbulence of this air stream is still capable of producing marked effects on transition.

The foregoing comparisons show the important effects of small amounts of turbulence. The need for similar tests for comparative purposes on this airfoil in flight is obvious.

Turbulent Boundary Layer

The shape of the turbulent-boundary-layer profiles plotted logarithmically shows that the profiles follow neither a power law nor a logarithmic law. The entire problem of the effect of pressure gradient on the turbulent boundary layer needs further study.

Drag

The curve of the effect of scale on drag given in figure 3 shows that the drag of the N.A.C.A. 0012 airfoil is relatively insensitive to Reynolds Number. The values are in good agreement with those from the N.A.C.A. variable-density tunnel corrected to effective Reynolds Number (reference 12) except for the variable-density-tunnel point at a Reynolds Number of 450,000, the accuracy of which is doubtful. Although the critical values of N on this airfoil (fig. 23) seem to give a good indication of the turbulence level of the air stream, the drag of this section is insufficiently sensitive to be used for this purpose.

Skin Friction

The distribution of skin friction along the surface of the airfoil was determined from the boundary-layer surveys. In the laminar-flow region, the skin friction was found from the slope of the velocity profiles at the surface. In terms of the parameters used in figures 4 to 10, the skin-friction coefficient is given by

$$c_f = \frac{2}{\sqrt{R}} \left(\frac{d\,\frac{u}{U_o}}{d\frac{y}{c}\sqrt{R}} \right)_{y=o}$$

Throughout the region of turbulent flow, the Von Kármán momentum relation was used to find the distribution of skin friction. In terms of the quantities measured in the tests, the skin-friction coefficient may be expressed as

$$c_f = 2\left\{ \frac{d}{ds}\left[\int_o^\infty \frac{u}{U_o}\left(\frac{U-u}{U_o}\right) dy \right] + \frac{d\left(\frac{U}{U_o}\right)}{ds} \int_o^\infty \left(\frac{U-u}{U_o}\right) dy \right\}$$

The distribution of skin friction in the turbulent layer and the profile drag of the airfoil were also calculated according to the method of Squire and Young (reference 3). In these theoretical calculations, transition was assumed to occur at the positions given in figure 22.

Curves showing the distribution of skin friction along the surface are given in figure 24. The integrated skin-friction drag coefficients are plotted in figure 2 for comparison with the results of the wake-survey measurements.

Figure 2 shows the drag coefficients calculated according to the method of reference 3 and those obtained from the wake-survey determinations to be in good agreement. The fact that the calculated values of the profile-drag coefficients are slightly higher than the experimental values is probably due to the choice of the position of the transition point. The transition point was considered to correspond to a point near the beginning of the transition region, where the skin friction is still low. If a point slightly farther back had been chosen, the agreement would have been better.

The theoretical and the experimental distributions of skin friction along the surface (fig. 24) are considered to be in good agreement, in view of the difficulty of determining the experimental distribution that involves the differentiation of an experimental curve. The agreement between the experimental and the theoretical integrated skin-friction coefficients (fig. 2) is much closer. The comparison of the integrated coefficients is regarded as a more reliable check of the theory than the curves of skin-friction distribution because of the greater accuracy in obtaining the integrated coefficients from the boundary-layer surveys. The distribution of velocities close to the surface of the airfoil (fig. 25) should give a fair indication of the shape of the skin-friction distributions. The shape of these curves is similar to that of the theoretical curves of skin friction. It is therefore thought that the Squire-and-Young method (reference 3) gives an accurate estimate of the turbulent skin-friction distribution, at least for moderate pressure gradients.

The theoretical calculations and the experimental results given in figure 2 are in agreement in showing that direct skin friction along the surface of this airfoil accounts for approximately 80 percent of the profile drag.

CONCLUSIONS

From the results of the present investigation, it is concluded that:

1. The calculated and the experimental laminar boundary-layer profiles for an N.A.C.A. 0012 airfoil are in good agreement.

2. Although comparative measurements of transition in flight indicate that the turbulence of the air stream of the low-turbulence tunnel is large enough to produce marked effects on transition, the turbulence of this air stream is less than that of other N.A.C.A. tunnels previously supposed to have low turbulence.

3. The critical Reynolds Number of a sphere cannot be used as a measure of the effects of small amounts of turbulence on the flow about an airfoil.

4. The calculated turbulent skin-friction distribution for an N.A.C.A. 0012 airfoil is in fair agreement with that found from boundary-layer surveys.

5. For the N.A.C.A. 0012 airfoil, the Squire-and-Young method of calculating profile drag gave results in good agreement with the value determined from wake surveys.

6. Approximately 80 percent of the profile drag of the N.A.C.A. 0012 airfoil is direct skin-friction drag.

Langley Memorial Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., June 7, 1940.

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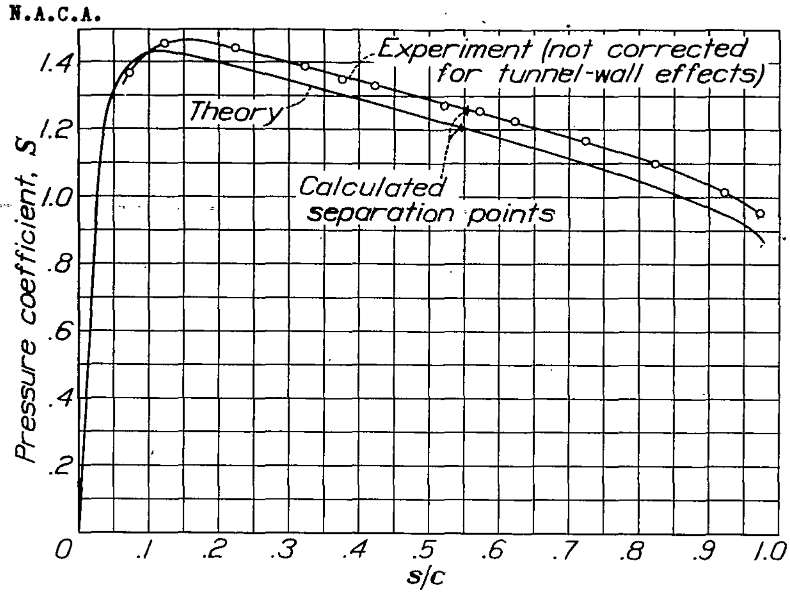

Figure 1.- Pressure distribution about N.A.C.A. 0012 airfoil.

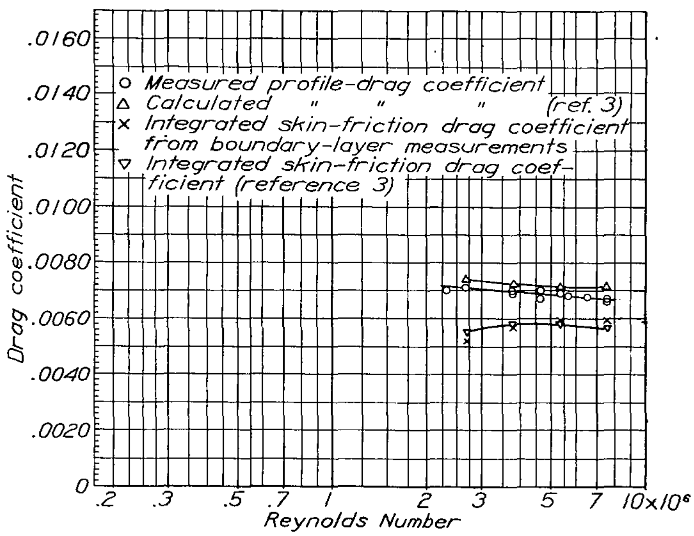

Figure 2.- Effect of scale on drag coefficient of 5-foot-chord N.A.C.A. 0012 airfoil.

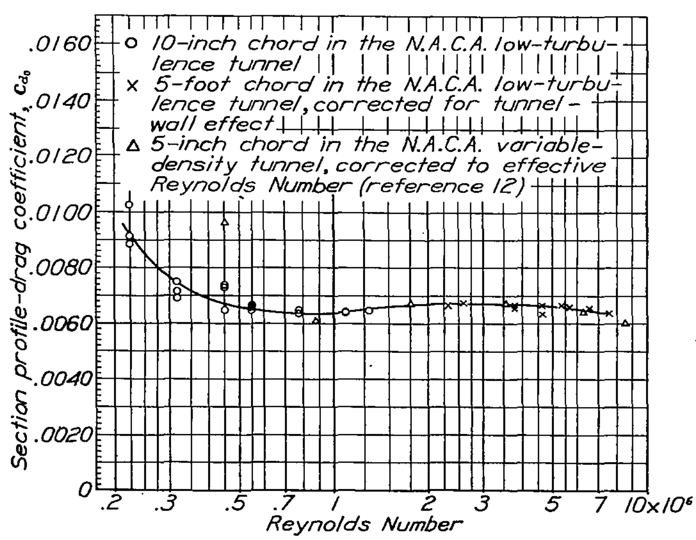

Figure 3.- Effect of scale on drag coefficient of N.A.C.A. 0012 airfoil.

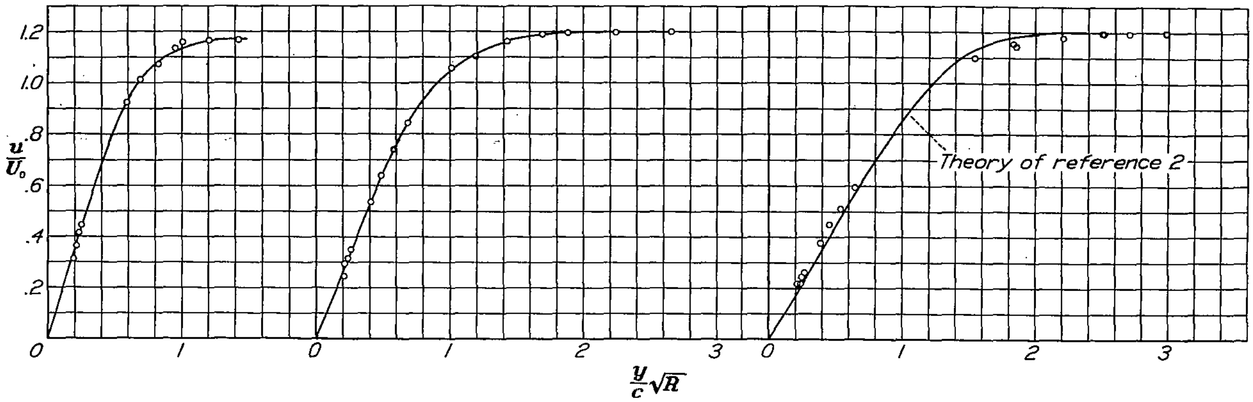

Figure 4.- Boundary-layer survey 0.07c along surface from leading edge of N.A.C.A. 0012 airfoil.

Figure 5.- Boundary-layer survey 0.12c along surface from leading edge of N.A.C.A. 0012 airfoil.

Figure 6.- Boundary-layer survey 0.22c along surface from leading edge of N.A.C.A. 0012 airfoil.

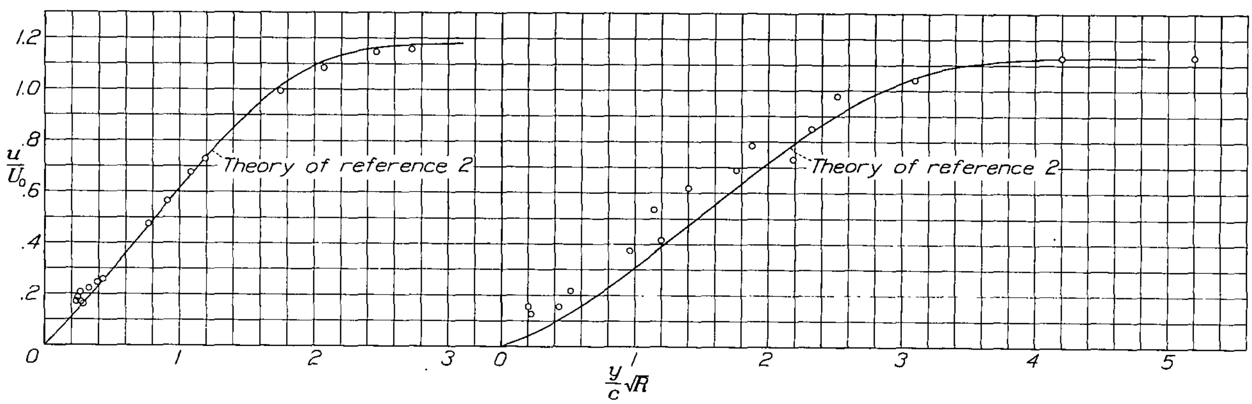

Figure 7.- Boundary-layer survey 0.32c along surface from leading edge of N.A.C.A. 0012 airfoil.

Figure 10.- Boundary-layer survey 0.52c along surface from leading edge of N.A.C.A. 0012 airfoil.

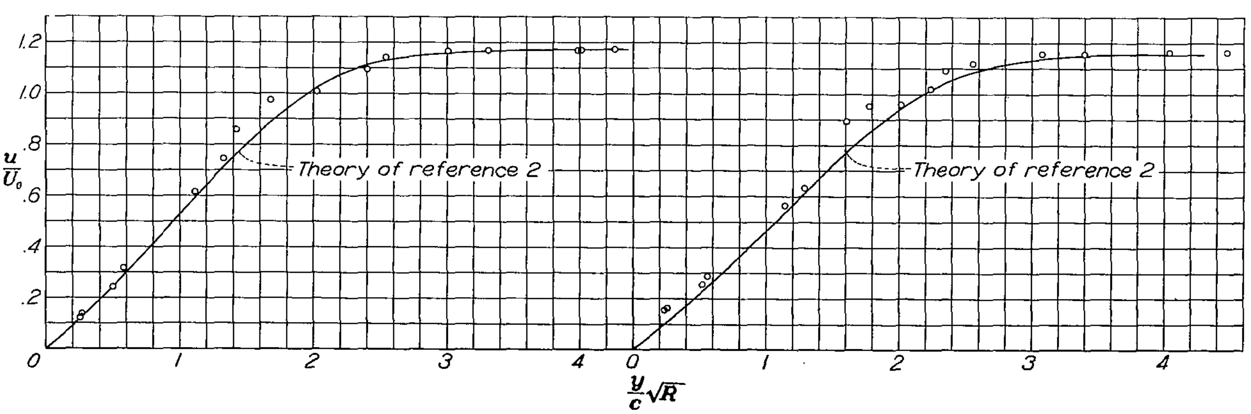

Figure 8.- Boundary-layer survey 0.37c along surface from leading edge of N.A.C.A. 0012 airfoil.

Figure 9.- Boundary-layer survey 0.42c along surface from leading edge of N.A.C.A. 0012 airfoil.

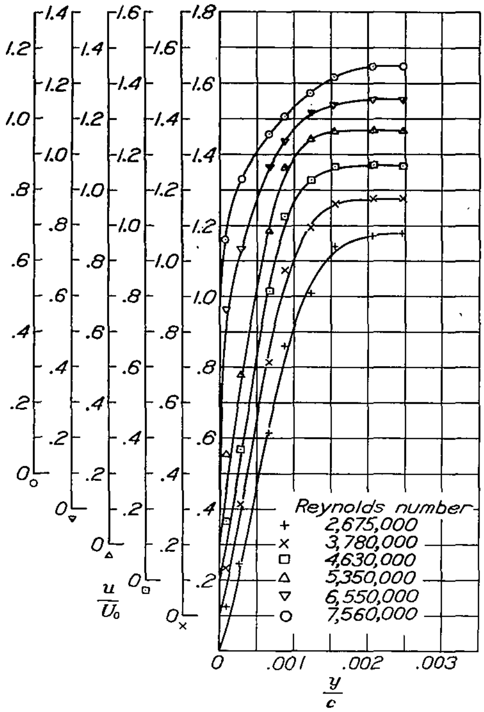

Figure 11.- Boundary-layer surveys 0.37c along surface from leading edge of N.A.C.A. 0012 airfoil.

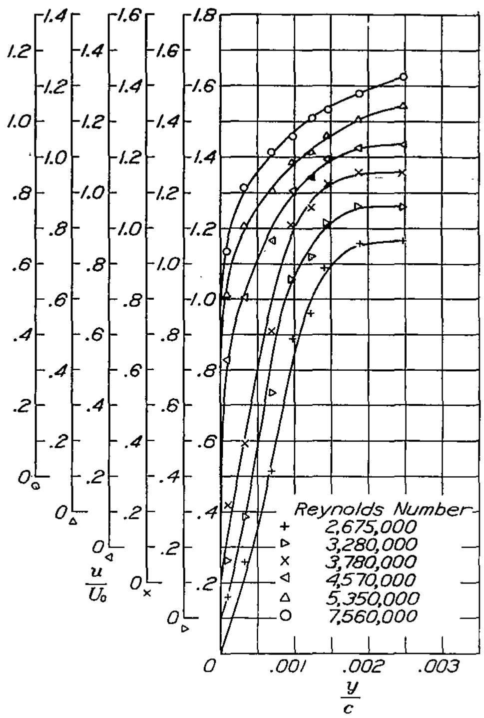

Figure 12.- Boundary-layer surveys 0.42c along surface from leading edge of N.A.C.A. 0012 airfoil.

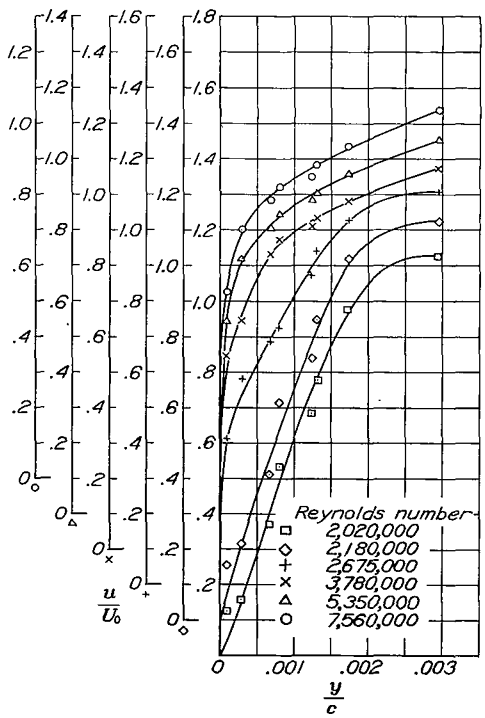

Figure 13.- Boundary-layer surveys 0.52c along surface from leading edge of N.A.C.A. 0012 airfoil.

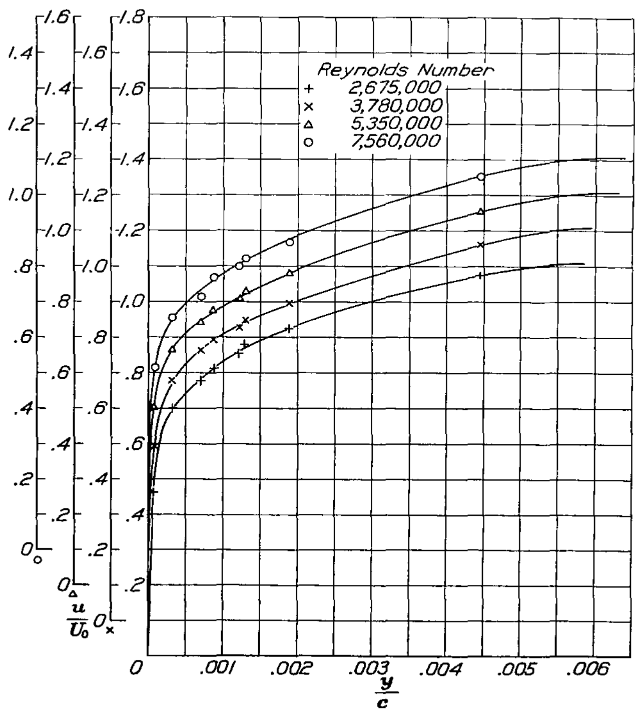

Figure 14.- Boundary-layer surveys 0.62c along surface from leading edge of N.A.C.A. 0012 airfoil.

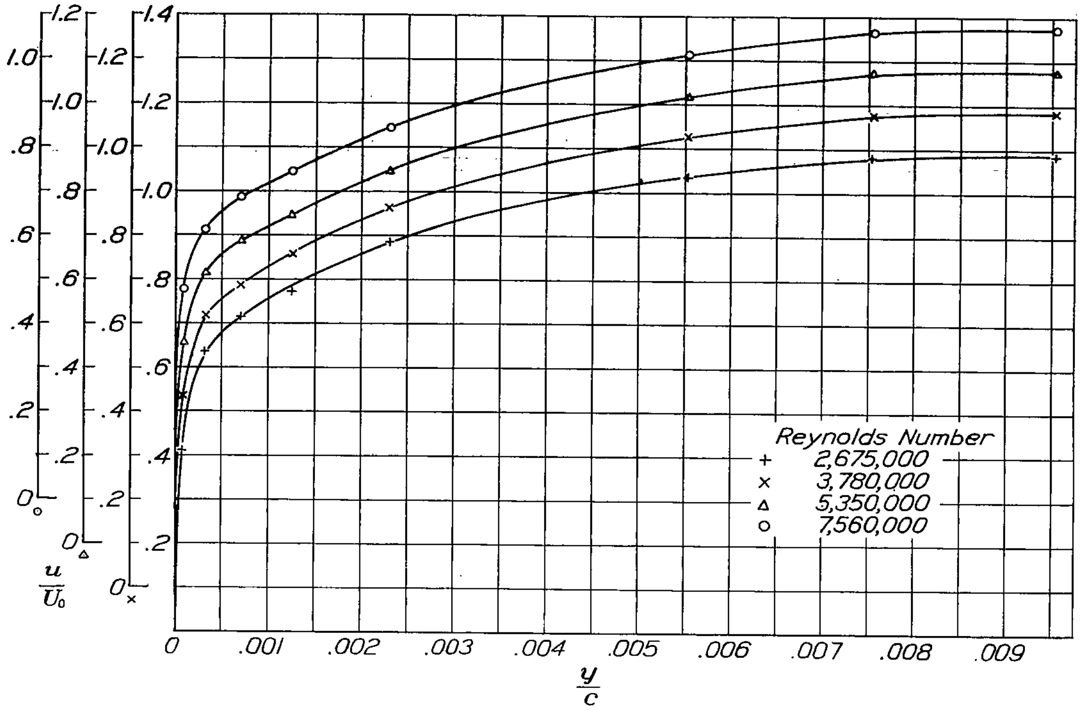

Figure 15.- Boundary-layer surveys 0.72c along surface from leading edge of N.A.C.A. 0012 airfoil.

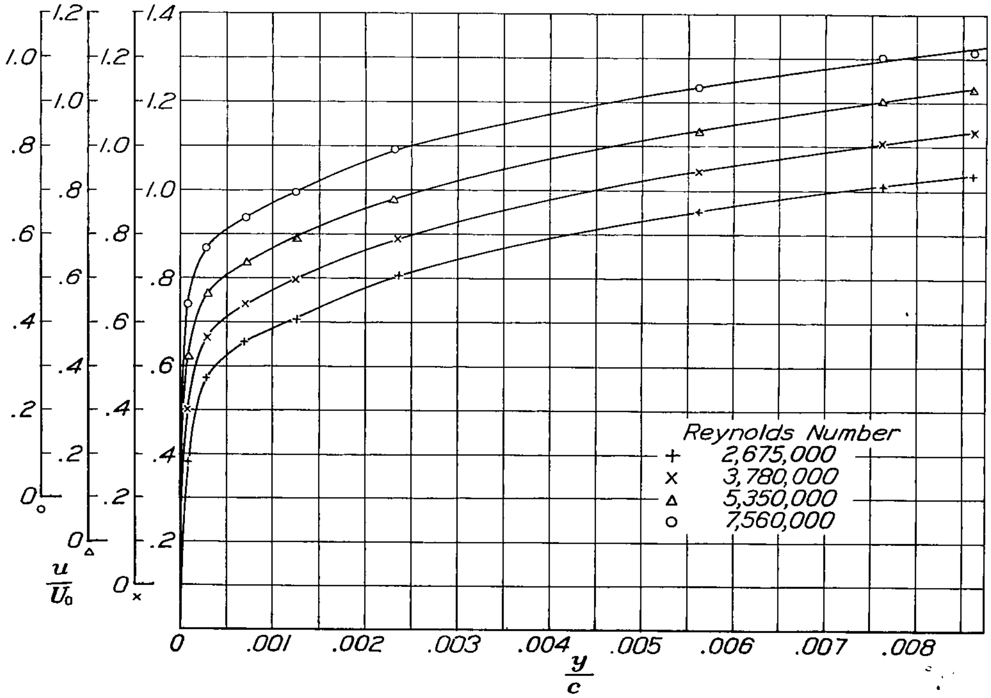

Figure 16.- Boundary-layer surveys 0.82c along surface from leading edge of N.A.C.A. 0012 airfoil.

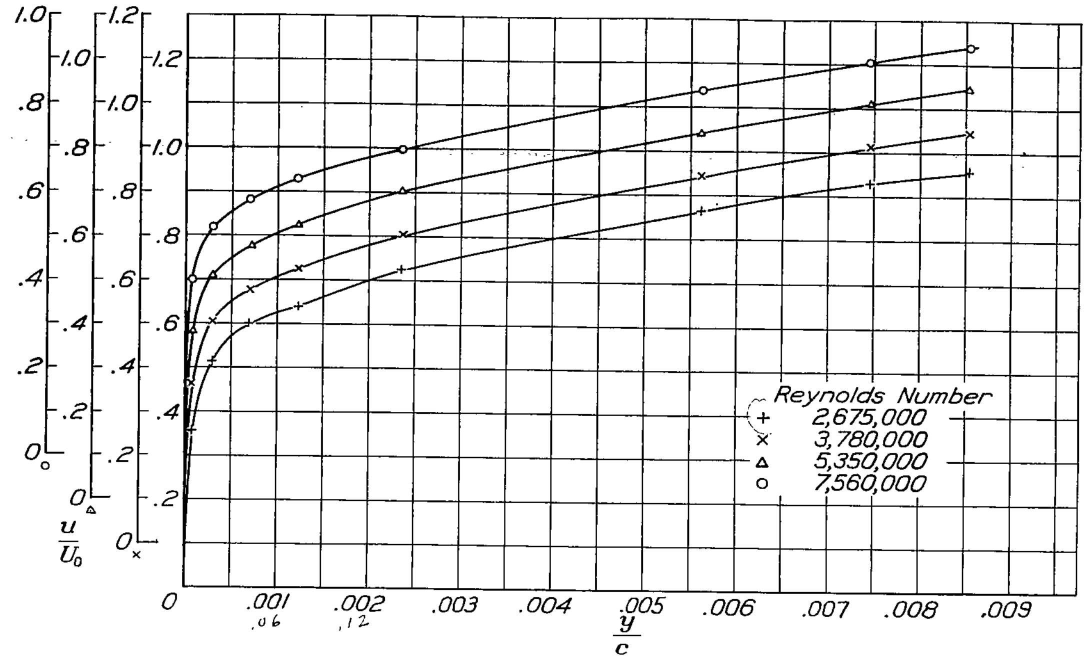

Figure 17.- Boundary-layer surveys 0.92c along surface from leading edge of N.A.C.A. 0012 airfoil.

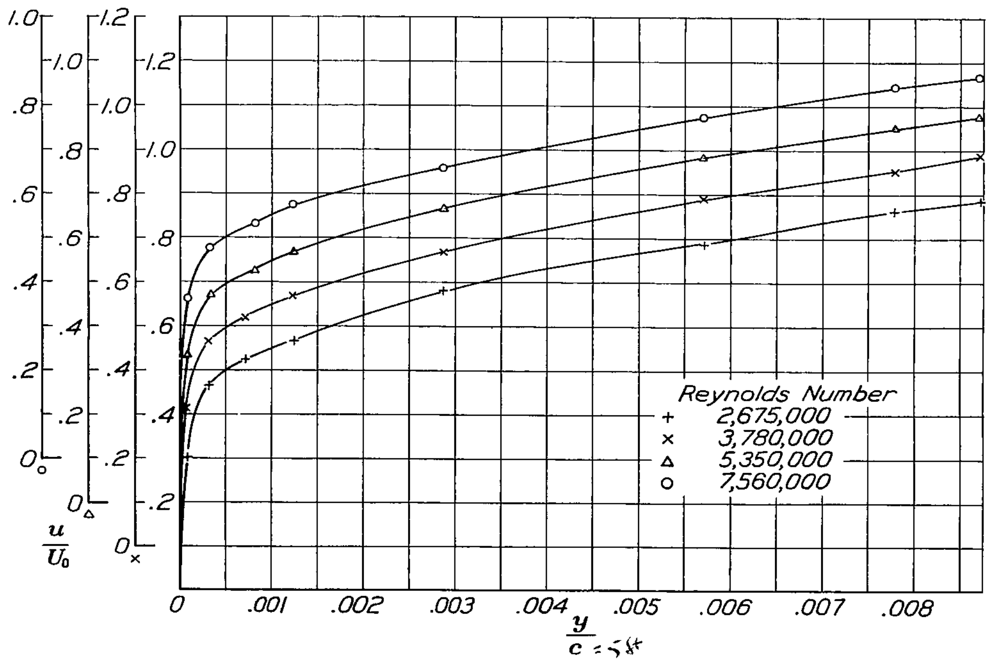

Figure 18.- Boundary-layer surveys 0.97c along surface from leading edge of N.A.C.A. 0012 airfoil.

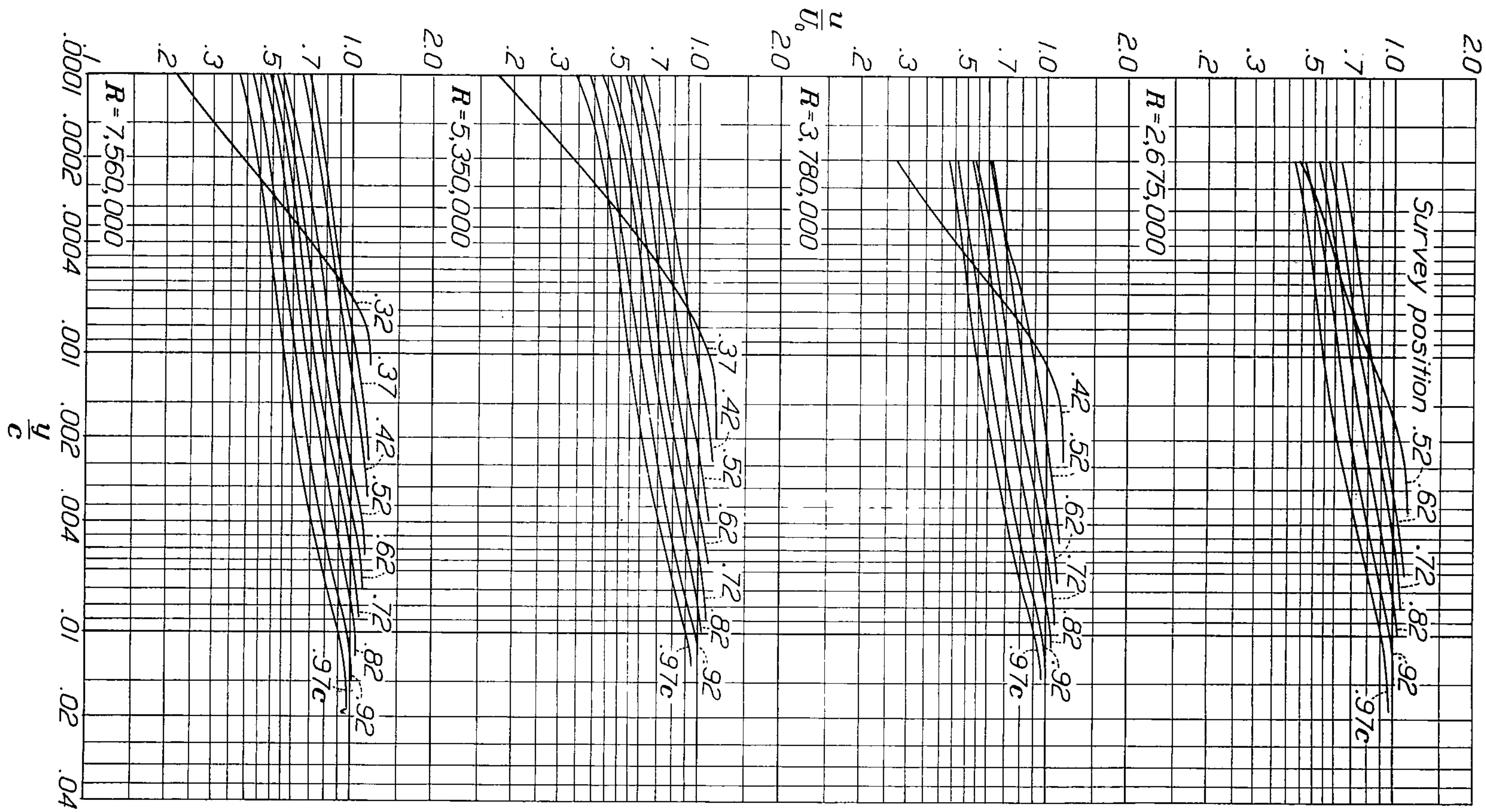

Figure 19.- Logarithmic plot of boundary-layer profiles along surface of N.A.C.A. 0012 airfoil.

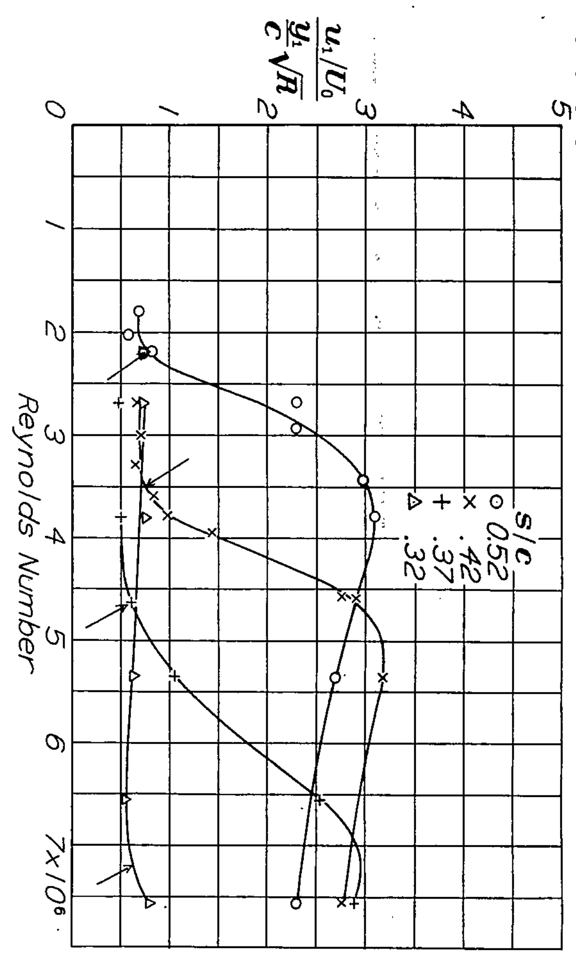

Figure 20.- Transition parameter as a function of the Reynolds Number for N.A.C.A. 0012 airfoil.

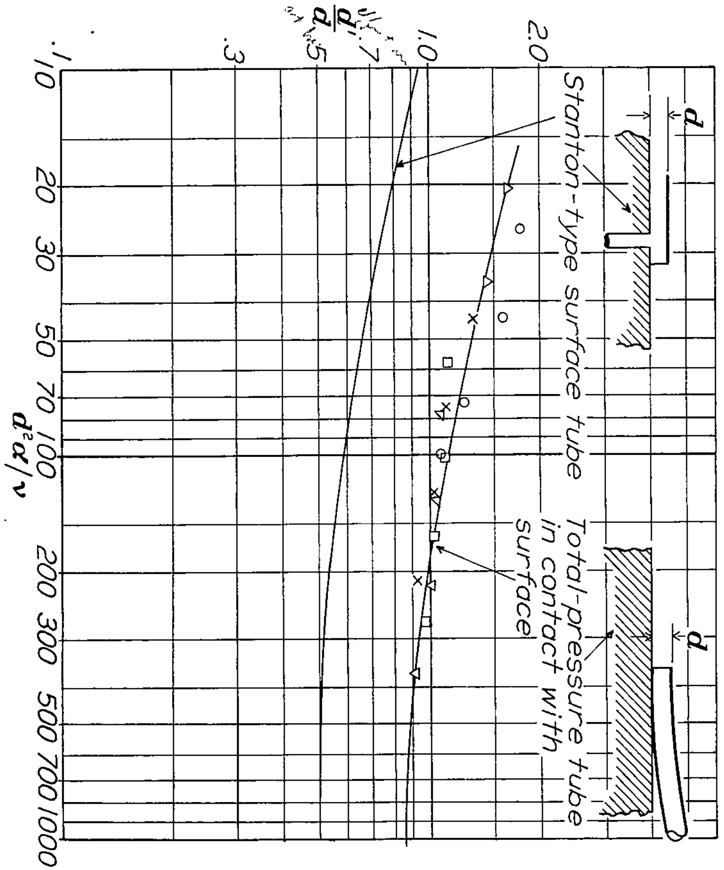

Figure 21.- Curves giving the effective height of Stanton-type surface tubes and total-pressure tubes in contact with the surface.

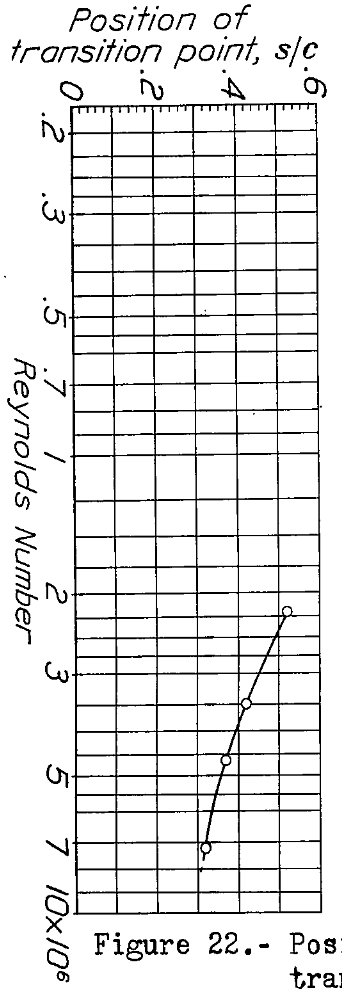

Figure 22.- Position of transition point as a function of Reynolds Number for N.A.C.A. 0012 airfoil.

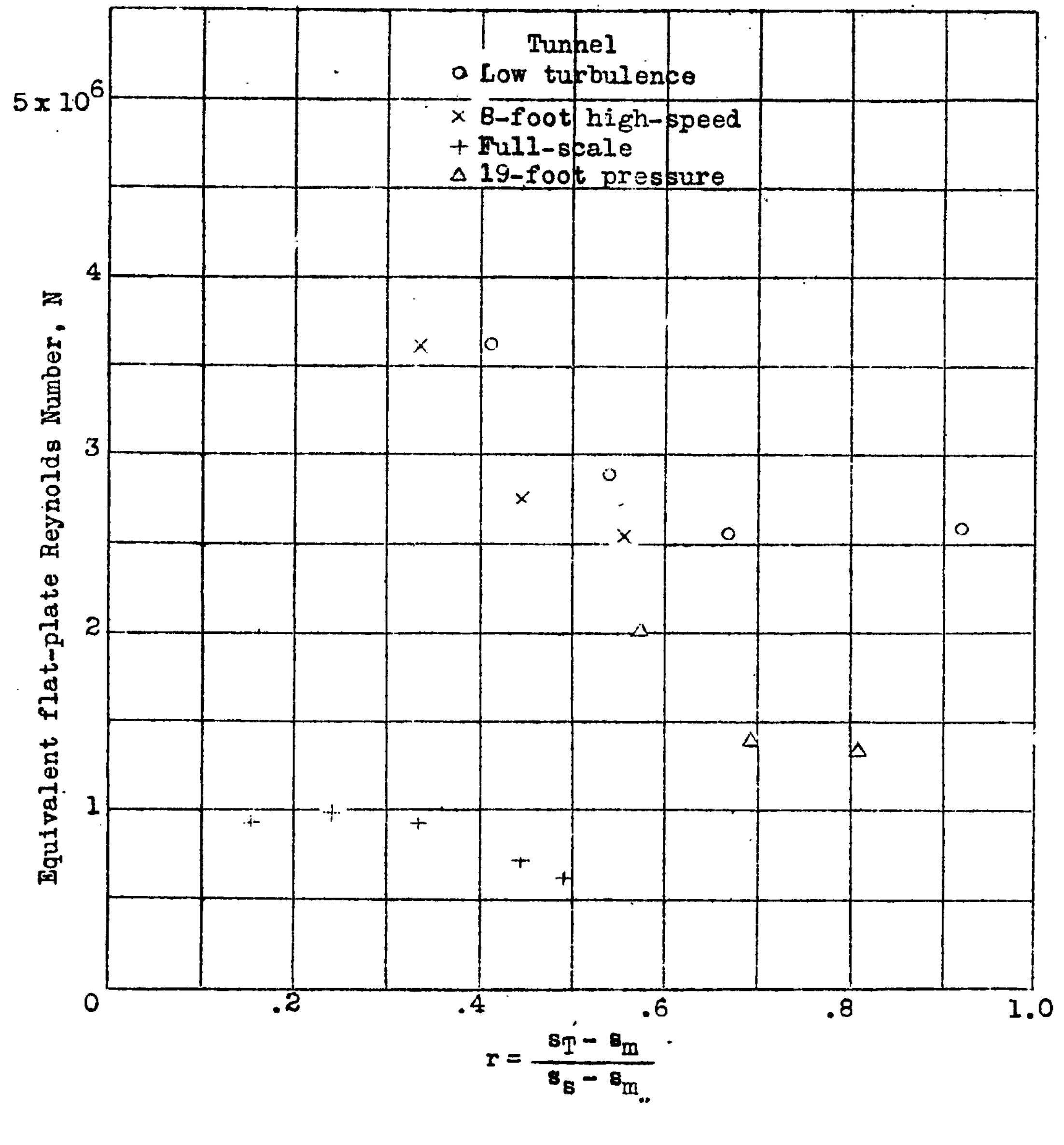

Figure 23.- Equivalent flat-plate Reynolds Numbers corresponding to transition on N.A.C.A. 0012 airfoil from tests in several N.A.C.A. tunnels.

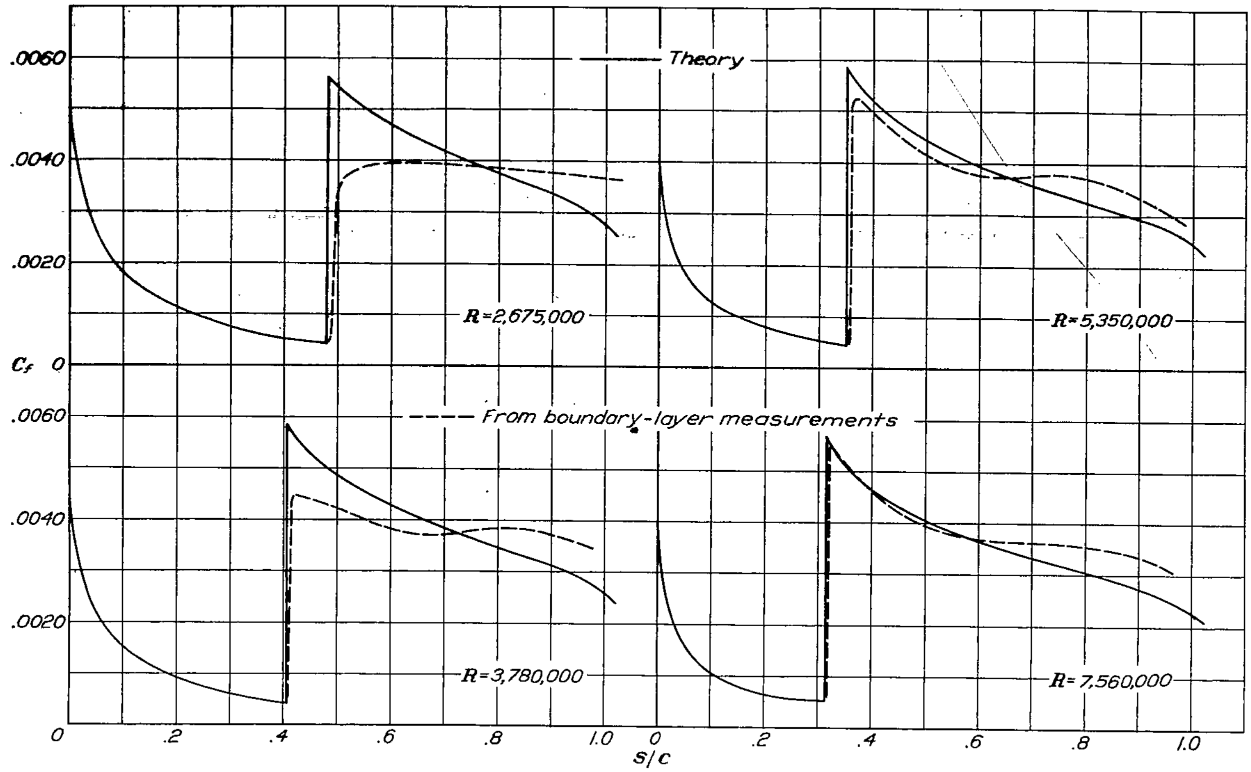

Figure 24.- Distribution of skin friction along the surface of N.A.C.A. 0012 airfoil.

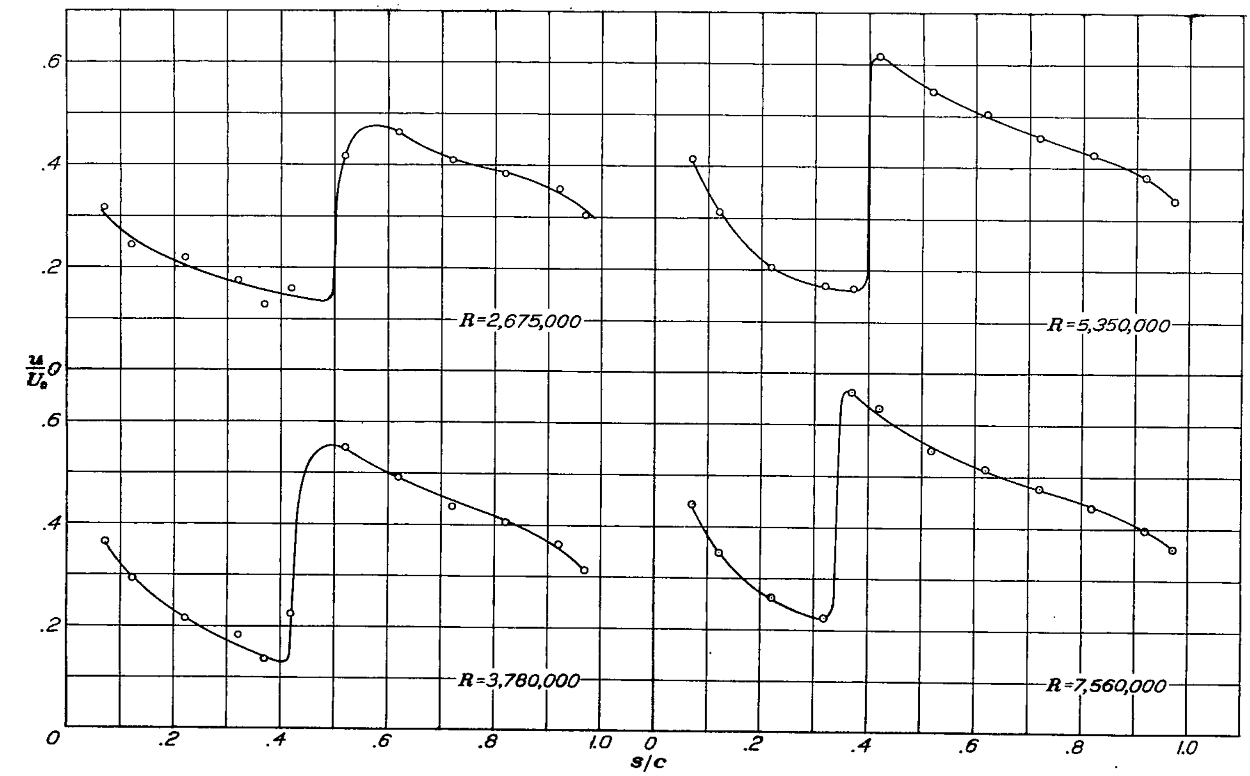

Figure 25.- Chordwise distribution of velocity near the surface of N.A.C.A. 0012 airfoil.